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STATUS REPORT ON THE LIQUID-SOLID ROCKE[illegible]

by

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ABSTRACT

A summary is presented of the initial phases of the research and development of a novel rocket propulsion system which combines both liquid and solid rocket components into a single liquid-solid rocket.

A liquid fuel, such as gasoline or kerosene, is injected into a chamber containing a solid inorganic oxidizer, such as ammonium nitrate, ammonium perchlorate or potassium perchlorate, the surface of which has been heated to its decomposition temperature. The resultant reaction raises the chamber pressure and temperature, and the hot gases are expelled through a nozzle to provide the desired thrust. In the course of an experimental program, various inorganic oxidizer compositions have been successfully fabricated and small-scale liquid-solid rockets have been tested sufficiently to demonstrate their feasibility.

Solid inorganic oxidizers were fabricated by temperature-compression molding processes and were tested in a small-scale rocket chamber using solvent naphtha as fuel. Test results for a potassium perchlorate composition showed specific impulse values of 50 per cent of those which can be obtained from solid rockets in which the fuel-oxidizer ratio is stoichiometric. Test results for other compositions showed specific impulse values as high as 70 per cent of the stoichiometric solid rocket value. The effect on the combustion chamber pressure of changes in oxidizer composition, catalyst, grain length, and ratio of surface area to throat area were investigated.

The results of this study indicate that the combined liquid-solid rocket is feasible, and that it shows promise of development into a reliable and regulatable propulsion unit which will give performance comparable with that of liquid or solid rockets.

Such a liquid-solid rocket propellant system has several advantages. The fuel and the oxidizer are separated from each other until they are needed for the reaction. The chemical materials used in the rocket are non-toxic, non-corrosive and non-explosive and thus the handling and manufacture of the rocket components can be accomplished in relative safety. Furthermore, the chemicals needed in the production of the rocket are abundant and can be obtained at low cost.

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I. Principal Symbols

A	area, square inches
B	experimental coefficient in chamber pressure-burning rate relation
C	coefficient
F	thrust, pounds
I	impulse
K	ratio of initial surface area of grain exposed for burning to throat area of exhaust nozzle
L	length, inches
M	average molecular weight
P	pressure, $lb/in.^2$ gauge
R	universal gas constant
S	surface area of the oxidizer
T	temperature
W	weight, pounds
d	diameter, inches
g	gravitational constant
r	linear burning rate, in./sec
t	time, seconds
α	ratio of oxygen to oxidizer
γ	ratio of specific heats
ϕ	equivalence ratio, $\frac{\text{oxidizer to fuel weight ratio, stoichiometric}}{\text{oxidizer to fuel weight ratio, actual}}$
ρ	weight density of the solid oxidizer

II. Subscript Symbols

D	discharge
F	thrust
b	burning
c	chamber
f	fuel
o	oxidizer
sp	specific
t	throat of the exhaust nozzle

III. Superscript Symbols

n	exponent in chamber pressure-burning rate relation
.	time rate of change
—	average

STATUS REPORT ON THE LIQUID-SOLID ROCKET[1]

I. INTRODUCTION

Purpose

The purpose of this study has been to evaluate the possibility of developing a new type of rocket which uses a liquid hydrocarbon fuel and a solid oxidizer.

The development of a liquid-solid rocket was undertaken in an attempt to provide an operational rocket propellant system which could be produced safely, quickly, and in quantity from materials which are in plentiful supply. In spite of the fact that rockets are currently being used more and more extensively as boosters, as propellants for missiles, and as assist-take-off devices, none of the propellant systems so far proposed satisfies all of the production requirements mentioned above. Of the three possible propellant systems - gas, liquid, and solid - the gas system is impracticable because of the poor performance which is the result of its weight and volume. The liquid propellant systems have the advantage in controllability, low cost, and potentially high performance, but they have some undesirable properties in common. All liquid oxidizers present severe storage and handling problems: some are corrosive such as nitric acid, some are toxic such as nitrogen tetroxide, and others such as liquid oxygen require insulation and refrigeration.

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[1] In this report the term "liquid-solid rocket" will be used to refer to a rocket which uses a liquid fuel and a solid oxidizer and the term "solid-liquid rocket" will be used to refer to a rocket which uses a solid fuel and a liquid oxidizer.

All require complex feed systems. In addition, precise control of fuel injection is required to prevent explosion hazards.

Solid propellants, too, are classified as explosives and must be processed, stored, and handled with due regard to safety precautions. The explosive hazard and the consequent safety measures necessary in processing add to their high cost. Although the solid propellant rocket is non-regulatable, it is reliable, simple, and applicable to short duration functions.

It is here suggested that the benefits of both liquid and solid propulsion systems may be combined in a liquid-solid rocket that is inexpensive, reliable, regulatable, relatively safe, using cheap, readily available chemicals, yet giving performance comparable to both liquid and solid rockets.

Historical Background

The development of rockets made great strides during World War II but little or no effort was directed toward developing liquid-solid rockets using solid oxidizers. Several studies were made by the Germans directed toward the utilization of the complementary system - solid fuels and liquid oxidizers. One such proposal for a solid-liquid rocket was suggested as early as 1933 by Sanger who considered employing aluminum powder suspended in a liquid hydrocarbon reacting with liquid oxygen (Ref. 1).[2] Later German experimenters studying the 'lithergol' propellants attempted to use nitrous oxide (N_2O) as a liquid oxidizer reacting with carbon as the solid fuel (Ref. 2).

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[2]All references appear on pages 45-47 of this report.

Recent American experimenters have followed the German example by using hydrogen peroxide as the liquid oxidizer which reacts with polythene fuel, a solid plastic (Ref. 3).

In England, solid oxidizers were developed independently in an attempt to provide auxiliary combustion agents for fuel-rich solid propellants. These oxidizers were included in a rocket chamber containing a double base (nitroglycerine, nitrocellulose) powder for the purpose of increasing the combustion efficiency, but were never employed with liquid fuels (Ref. 4)

A related program of research in England and the United States dealt with the development of plastic propellants in which a solid oxidizer is held in a matrix of an organic gel binder which contains oxidizable material. Propellants of this type were classified as composite propellants and were developed extensively by OSRD during World War II (Refs. 5, 6). These propellants were merely composition variations of standard solid rocket propellants. Studies on other related problems were undertaken in an attempt to utilize the nitrates and perchlorates as the oxidizing agents for the asphalt, resin, or plastic type composites in which they are imbedded (Refs. 7, 8, 9, 10, 11). All of these studies treat both the fuel and the oxidizer as solid constituents of a solid rocket propellant.

There is no evidence in the literature of any previous attempts to combine a liquid hydrocarbon fuel with a solid inorganic oxidizer as a rocket propulsion system.

Comparison of Rocket Propellant Systems

Rockets are generally classified according to the state of the stored material which is used for propulsion. Table I is a comparison of the various propellant systems with respect to a set of criteria which has been generally agreed upon as significant (Refs. 12, 13, 14, 15, 16).

TABLE I

Comparison of Rocket Propellant Systems

System	Gas	Liquid (Ref. 17)		Solid (Ref. 18)	Liquid-Solid
Example	CO_2	Alcohol - Liquid Oxygen	JP3 - Nitric Acid	Double Base	Naphtha - NH_4NO_3 $KClO_4$
A. Supply					
1. Availability	Unlimited	Unlimited	Unlimited	Limited	Wide
2. Source	Limestone	Petroleum by-products Air	Petroleum Ammonia oxidation	Cellulose; Glycerine; Nitric Acid	Petroleum Ammonia
3. Cost	High a/c packaging	Fair	Fair	High a/c processing	Low
B. Performance					
1. Impulse-weight ratio	Low	Fair	Fair	Good	Fair
2. Specific impulse	Low	High (240)	High (218)	Fair (200)	Fair
C. Safety					
1. Explosive	No	Mild	Mild	Yes, during processing	No
2. Corrosive	No	Slight	Very	Yes, during processing	No
3. Inflammable	No	High	Dangerous	Yes	Slight
4. Toxic	No	Yes	Very	Mild (fumes)	No
D. Operation					
1. Simple	Yes	No	No	Yes	Yes
2. Controllable	No	Yes	Yes	No	Yes
3. Reliable	Yes	No	No	Yes	Yes
4. Storable	Yes	Difficult	Difficult	Within temperature range	Yes
Applications	Models, Toys	Guided missiles; long range projectiles		ATO; artillery; display	ATO; ram boost
Main function	Low, brief thrusts	Extended operation		High, brief thrusts	Controlled, high thrusts

II. GENERAL DESCRIPTION OF THE LIQUID-SOLID ROCKET

The liquid-solid rocket propulsion system proposed here operates in the following manner. An oxidizer composed of a mixture of inorganic oxidizer salts, which are commercially available and which can be stored, handled, and processed as ordinary chemicals, is fabricated into a grain of the proper size and configuration by conventional chemical engineering methods. The solid oxidizer is stable, non-explosive, and, by itself, non-inflammable. It resides within a rocket chamber of an appropriate design. Connected to the rocket chamber is an external fuel-supply line from the separately-stored liquid fuel (such as an airplane fuel tank, if the rocket is to be used to assist take-off).

To operate the rocket, an igniter system within the rocket chamber is turned on. The igniter system provides a source of intense heat which sharply increases the surface temperature of the oxidizer. The oxidizer begins to decompose, thereby releasing free oxygen from its surface. The hydrocarbon fuel is then introduced under pressure. The resultant reaction between hydrocarbon and free oxygen raises the temperature and pressure of the gas within the rocket chamber, and the combustion products are discharged through a nozzle to produce the forward thrust.

The flame temperature is high enough to maintain a decomposition reaction at the solid oxidizer surface which thus supplies more oxygen for the reaction. This additional oxygen reacts with more hydrocarbon fuel and the reaction continues until a steady state is attained in which the rate of production of combustion gases is equal to the rate of discharge through the nozzle.

The reaction may be continued until all the solid oxidizer is consumed or the reaction may be quenched by turning off the fuel flow. If the reaction is interrupted, the remaining solid oxidizer can be reignited and used at some later time. The process described above can be repeated until all the oxidizer has been consumed.

From this description, it may be seen that the proposed propulsion scheme has two specific problems:

1. the fabrication of an available oxidizer which will release oxygen readily upon heating, yet which will be non-explosive, inexpensive, non-toxic, non-inflammable, stable, and capable of being safely formed into the desired size and shape; and
2. the testing of the oxidizer grain with a liquid fuel to establish a combustion reaction, a suitable ignition system and fuel injection method, and to measure the critical rocket variables such as burning rate, fuel flow, pressures, and thrusts.

III. PROCEDURE

Theoretical

Before undertaking an intensive experimental program on the liquid-solid system, it was first necessary to confirm the chemical principles underlying the combustion reaction. A search of the literature was made to evaluate the various available solid oxidizers. Table II lists some of the important properties of the oxidizers considered and compares them with some liquid oxidizers.

Some preliminary calculations were made to determine whether the temperature rise in the reaction products would be sufficiently high to maintain a decomposition reaction at the oxidizer surface. It was found that a sustaining reaction would exist provided the depth of thermal penetration were small. In an oxidizer which has relatively low thermal conductivity, the heat of the reaction would go into raising the temperature of the oxidizer surface rather than into raising the temperature of the entire oxidizer mass.

These calculations were verified in the initial bench tests. Powdered potassium perchlorate was heated until oxygen was evolved, and a stream of fuel gas was directed at the surface as the heat source was removed. A bright pink flame continued to burn so long as the fuel was directed against the decomposing surface and the oxygen was not depleted.

Other studies were made to determine how the theory of liquid and of solid rockets could be combined for use in the design of a liquid-solid rocket. Unlike the liquid rocket system, in which the mixture ratio may be selected or varied by separate control of both oxidizer and fuel, the liquid-solid

TABLE II

Oxidizer Properties

Chemical	Formula	Molecular Weight	Specific Gravity	State	Melting Point (°C)	Boiling Point (°C)	Decomposition Point (°C)	% Oxygen Available (by weight)	Cost* (Ref. 18)
Hydrogen peroxide (dilute)	H_2O_2 H_2O	52.04	1.23	Liquid		100		30.8	
Nitrous oxide	N_2O	44.02	1.23	Liquid	-102.4	-88.5		36.4	
Potassium perchlorate	$KClO_4$	138.55	2.52	Solid	610 ±10		625	46.2	0.14 to 0.18¢/lb
Potassium nitrate	KNO_2	101.1	2.11	Solid	334		400	48.8	0.10 to 0.18¢/lb
Tetra nitro methane	$C(NO_2)_4$	196.02	1.65	Liquid	12	126.7		49.0	
Sodium perchlorate	$NaClO_4$	122.48		Solid			482	52.3	
Ammonium perchlorate	NH_4ClO_4	117.5	1.95	Solid			130	54.4	
Sodium nitrate	$NaNO_2$	55.01	2.26	Solid	307		380	27.0	0.023 to 0.026/lb
Nitrosyl perchlorate (Ref. 20)	$NOClO_4$	129.5	2.10	Solid			~100	58.8	
Ammonium nitrate	NH_4NO_2	80.05	1.73	Solid	169.6		210	60.0	0.025 to 0.036/lb
Nitrogen sesqui oxide	N_2O_2	75.02	1.45	Liquid	-102	2.5	2.8	52.2	
Perchloric acid	$HClO_4$	100.5	1.75	Liquid	-112	28 (50mm/Hg)		54.0	
Nitrogen dioxide	NO_2	46.01	1.48	Liquid	-8.2	21.2	21.3	88.8	
Nitrogen pentoxide	N_2O_5	108	1.64	Solid	30	47	47	74.2	
Nitric acid	HNO_2	52.0	1.50	Liquid	-42	56		78.3	0.04 to 0.15/lb
Hydrogen peroxide (conc.)	H_2O_2	34.02	1.46	Liquid	-1.7	152.1		94.2 conc.	0.20 to 0.25/lb
Liquid oxygen	O_2	32.0	1.14 (-152°C)	Liquid	-215.4	-182		100	
Ltqntd ozone	O_3	48.0	1.71 (-153°C)	Liquid	-251	-112		100	

*Cost is given in dollars/lb of ton lots where available.

rocket permits variation of the mixture ratio through control of the fuel flow only. Before the combustion in the liquid-solid rocket attains a steady state, the decomposition rate of the oxidizer depends upon the amount of heat absorbed during the ignition phase and upon the rate of injection of the fuel. Thus, if the rate of increase of the fuel flow is too rapid, the system may be flooded and the reaction quenched; while if the rate of increase of the fuel flow is too slow, the oxidizer may exhaust its supply of available oxygen without ever attaining a chamber pressure large enough to produce a practical thrust. This means that variations in the ignition and fuel flow may change the operating pressure and consequently may vary the thrust produced.

Unlike the solid rocket system, in which a constant composition is present throughout the burning period, the liquid-solid rocket has a varying composition until steady state is attained. Thus, the empirical burning-rate equation for solid rockets, $r = BP_c^n$ (Fig. 15), is applicable to only steady-state conditions in which the fuel flow rate is held constant and the decomposition rate of the oxidizer is assumed to be constant, thus providing a constant composition within the burning chamber.

The variables so far measured in the course of this investigation have been the solid oxidizer dimensions, length (L), outside diameter (d_1), inside diameter (d_2), and weights before (W_1) and after (W_2) burning; the liquid fuel flow rate ($\dot{W}_f$); the burning time (t_b); the area of the throat (A_t); the chamber pressure (P_c); and the thrust (F). From these basic measurements values of the following variables may be calculated according to the following equations:

$$(1) \quad r = \frac{d_1 - d_2}{2t_b} \qquad (5) \quad I_{sp} = \frac{F}{\dot{W}}$$

$$(2) \quad \dot{W}_o = \frac{W_1 - W_2}{t_b} \qquad (6) \quad I_{sp_o} = \frac{Ft_b}{W_1 - W_2}$$

$$(3) \quad \dot{W} = \dot{W}_o + \dot{W}_f \qquad (7) \quad C_F = \frac{F}{P_c A_t}$$

$$(4) \quad I = Ft_b \qquad (8) \quad K = \frac{S}{A_t}$$

where

r is the linear burning rate,

$\dot{W}_o$ is the weight flow rate of the oxidizer,

$\dot{W}$ is the weight rate of discharge

I is the total impulse,

I_{sp} is the specific impulse,

I_{sp_o} is the oxidizer specific impulse,

C_F is the thrust coefficient,

K is an arbitrary coefficient dependent upon oxidizer composition, and

S is the initial surface area of the oxidizer.

Now from liquid rocket theory we may use the following relations:

$$\dot{W}_o + \dot{W}_f = \dot{W} = C_D P_c A_t \qquad (9)$$

in which the weight discharge coefficient for choking conditions is given by

$$C_D = \sqrt{\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma + 1}{\gamma - 1}}} \cdot \sqrt{\frac{\gamma g}{\frac{R}{M} T_c}}$$

where

g is the gravitational constant,

γ is the ratio of the specific heats of the discharge gas composition,

R is the universal gas constant,

M is the average molecular weight of the discharge gas, and

T_c is the chamber gas temperature;

and

$$\phi = \frac{\text{stoichiometric oxygen/fuel}}{\text{actual oxygen/fuel}} = \frac{K_1}{\alpha(\dot{W}_o/\dot{W}_f)} \tag{10}$$

where α is the ratio of oxygen to oxidizer.

From solid rocket theory, we have

$$\dot{W}_o = \rho S r \tag{11}$$

where ρ is weight density.

Then, for the steady state of a liquid-solid rocket, we have

$$\rho S r = \frac{\dot{W}_f K_1}{\alpha \phi} \tag{12}$$

and

$$P_c = \frac{\dot{W}_o + \dot{W}_f}{C_D A_t} = \frac{\rho S r + \dot{W}_f}{C_D A_t} = \frac{\dot{W}_f \left(1 + \frac{K_1}{\alpha \phi}\right)}{C_D A_t} . \qquad (13)$$

From Eq. (12) it can be seen that

$$r = \frac{\dot{W}_f}{S} \left(\frac{K_1}{\rho \alpha \phi}\right). \qquad (14)$$

Since K_1, ρ, α are constants and since ϕ may be presumed to be nearly constant during burning, Eq. (14) becomes

$$r = \frac{\dot{W}_f}{S} (K_2) \qquad (15)$$

where $K_2 = \frac{K_1}{\rho \alpha \phi}$. Thus, the linear burning rate of an oxidizer may be determined empirically as a function of the ratio of the fuel flow rate and the oxidizer surface area, $\frac{\dot{W}_f}{S}$, for a given oxidizer composition.

The experimentally recorded chamber pressure may then be compared with the theoretical chamber pressure value found by inserting the measured value of r in Eq. (13). The measured thrusts may be compared with theoretical values obtained from the relation

$$F = C_F P_c A_t .$$

C_F can be optimized by an appropriate choice of chamber pressure and nozzle expansion area (Refs. 6, 14).

On the basis of calculations and preliminary tests which used the indicated combination of liquid and solid rocket theory an experimental program was undertaken.

Experimental

Fabrication of the Oxidizer

The basic problems in fabrication consist of (1) choosing an oxidizer, (2) determining its physical, chemical, and combustion properties, and (3) transforming the oxidizer into a suitable rocket grain.

Selection of an oxidizer---From the oxidizer properties listed in Table I the following three oxidizers were chosen for the preliminary studies:

(1) Potassium perchlorate
(2) Ammonium nitrate
(3) Ammonium perchlorate

Catalysts for increasing the decomposition rate, and binders for compounding the oxidizer powders were also investigated.

Determination of combustion properties---Bench tests using a Bunsen burner were then made on the selected oxidizers to determine whether they (1) evolved oxygen at a fast enough rate upon heating to sustain a combustion reaction, (2) left no solid residue, (3) quenched a self sustaining decomposition reaction

after removal of the heat source, and (4) sustained a combustion reaction when mixed with liquid fuel. The oxidizers were also tested for their sensitivity to detonation by the use of standard drop tests.

Transforming the oxidizer into a rocket grain---The oxidizer grain consists of various inorganic salts which have been ground to a fine powder, mixed in the proper proportions with appropriate catalysts and binders, placed in a mold of the desired size and shape, and pressed or cast into a rocket grain. For small-scale testing in a 2-inch diameter combustion chamber, perforated grains 5 to 6 inches long, 2 inches in outside diameter and 1 inch inside diameter were made. The rocket chamber was used as a mold in a combined casting and molding operation. The 2-inch diameter mold is shown in Fig. 1. Other methods of grain fabrication included the cold compression molding of test grains (Fig. 2), the casting of briquets for use in bench tests (Fig. 3), the split mold compression of loose grains for use in multiple charges, and the strip mold compression method which was tried in an attempt to simplify the process of removing the grain from the mold.

The method of fabrication adopted in any particular case depended upon the melting point of the oxidizer, the decomposition temperature of the oxidizer in the presence of its catalyst, the size and shape of the grain configuration, the number of grains required, and the use for which the grain was designed.

Testing

The basic problems in testing the liquid-solid rocket consist of (1) igniting the propellant system, (2) sustaining the combustion reaction, and (3) measuring the critical rocket

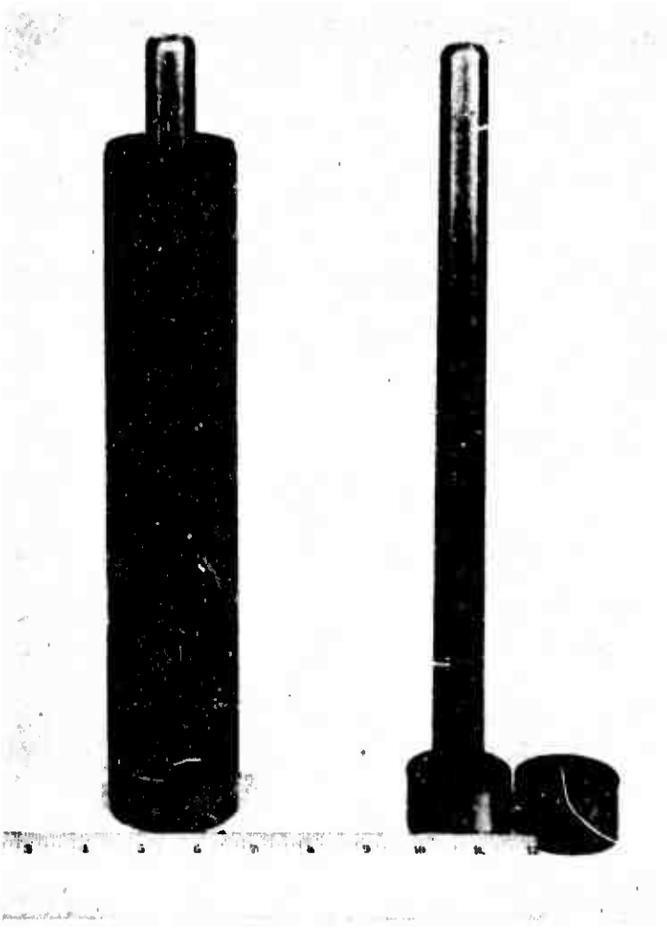


Fig. 1 MOTOR MOLD USED FOR FABRICATION OF PERFORATED GRAINS

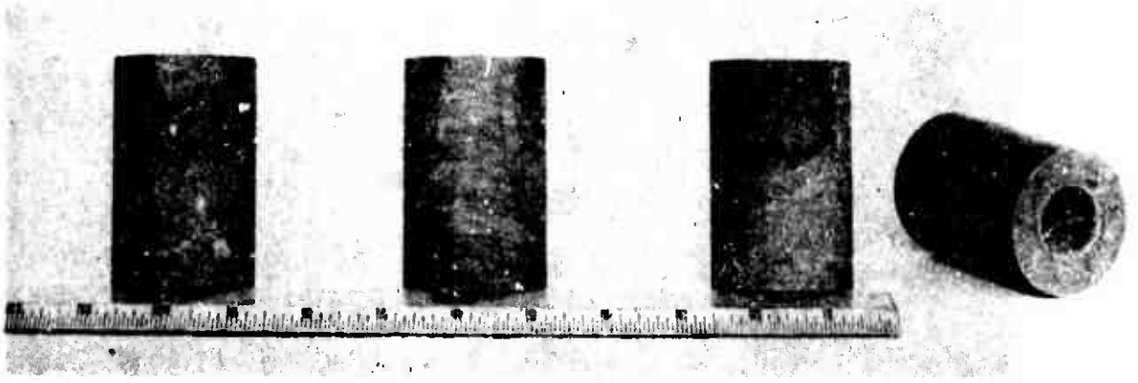


Fig. 2 COLD COMPRESSION MOLDED GRAINS

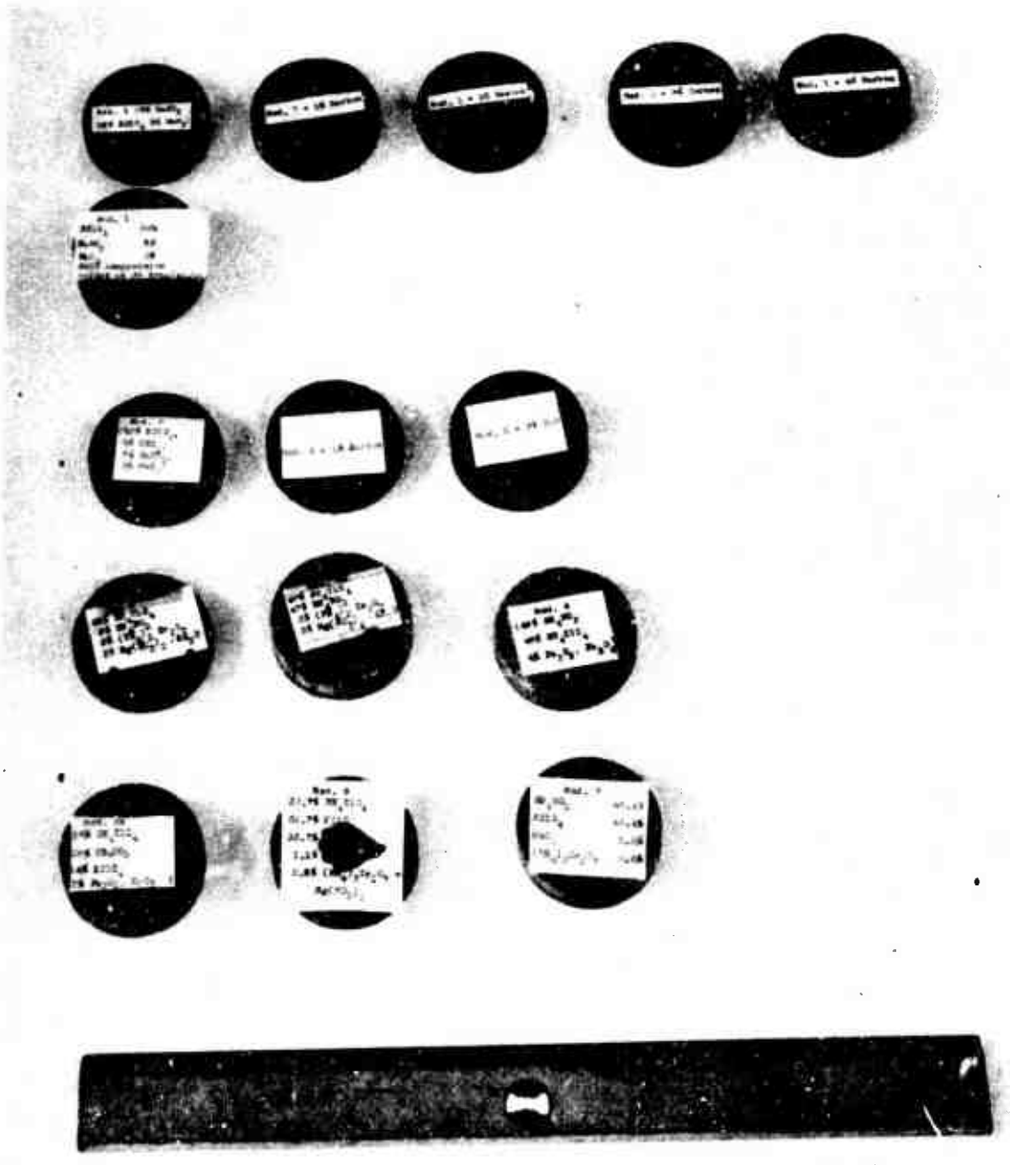


Fig. 3 OXIDIZER BRIQUETS

These briquets were fabricated by casting for use in bench tests.

performance values (Refs. 21, 22, 23). The purpose of these tests has been to obtain engineering data and to establish proper operating techniques for subsequent scale-up to full size JATO units.

The selected oxidizers and fuel were evaluated through static tests using 2-inch inside diameter thrust chambers. Measurements were made of the thrust, chamber pressure and fuel flow rate.

The 2-inch inside diameter thrust chamber in which the small-scale development testing was performed consisted of a head cap for fuel injection and ignition, a combustion chamber for housing the grain, and an exhaust nozzle assembly incorporating a shear ring for the purpose of increasing the safety of the assembly. The components of this thrust chamber are shown in Fig. 4.

On the basis of results obtained with the 2-inch thrust chamber, scale-up calculations were made for a 100-pound thrust chamber. The 6-inch inside diameter combustion chamber which was built on the basis of these calculations is shown assembled in Fig. 5 and in cross-section in Fig. 6. The design of full-scale liquid-solid JATO units will be guided by the scaling parameters determined in the 100-pound thrust unit tests.

The ignition system selected for heating the grain to its decomposition temperature was a combustible mixture of oxygen and solvent naphtha which was ignited by a spark in the head cap (Fig. 4). Several experiments indicated that a satisfactory ignition technique is a prerequisite for proper grain performance. Three methods of mixing the ignition materials were tried: (1) tangential, (2) perpendicular, and (3) angular. Figure 7 is a diagram of the igniters tested. Mod. 2, the angular mixer, was the igniter which provided the most effective heating of the oxidizer grain surface, for it directed the hot igniter gases into the perforation of the grain where the highest oxygen-evolution rate occurs.

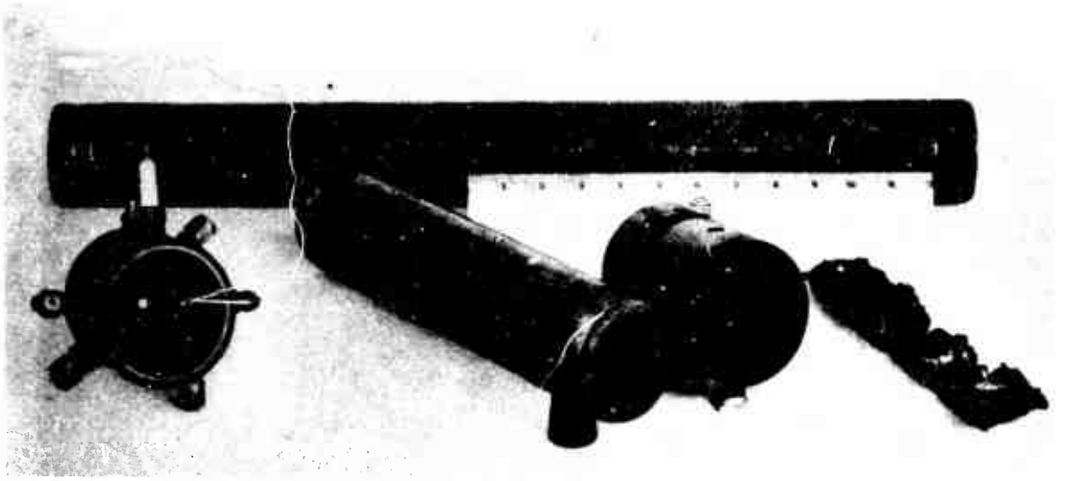


Fig. 4 TWO-INCH I.D. THRUST CHAMBER COMPONENTS

In the foreground (left to right) are shown the head cap, the 12-inch-long combustion chamber loaded with grain, the exhaust-nozzle assembly, and a typical Mod. 1 oxidizer residue. In the background is shown a 24-inch-long combustion chamber.

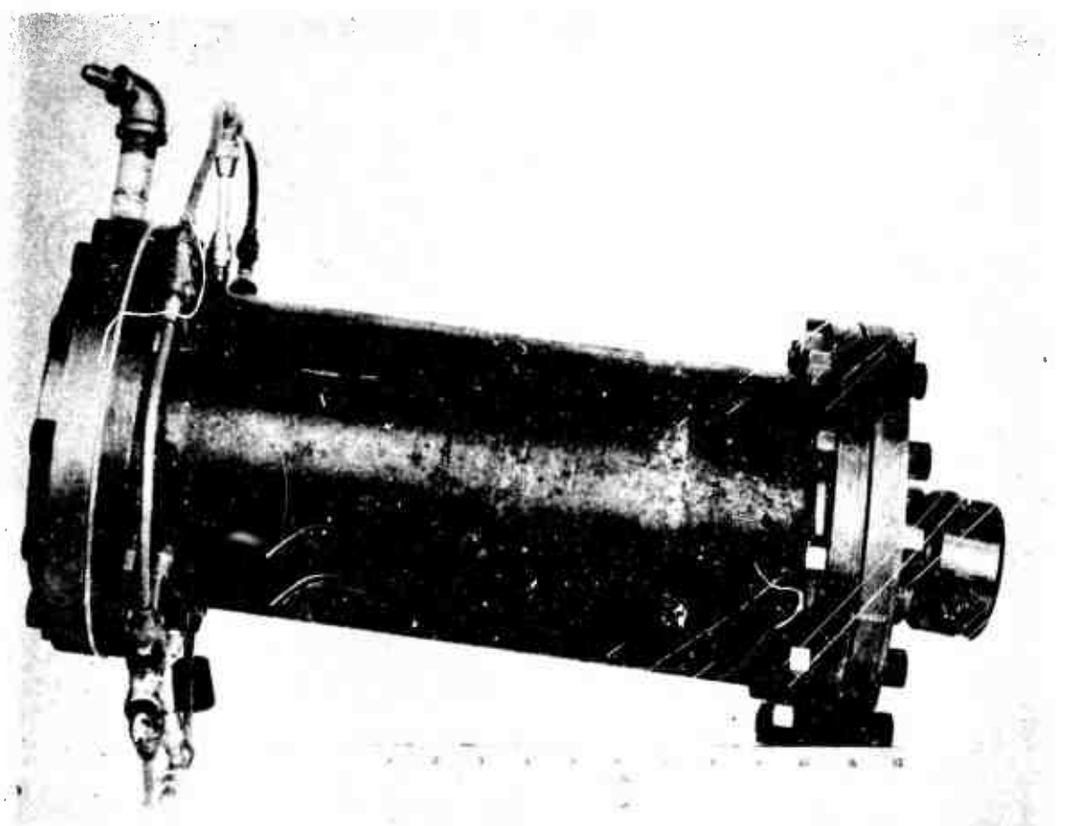


Fig. 5 SIX-INCH I.D. THRUST CHAMBER

The components of the thrust chamber (left to right) are the head cap with inlet to fuel manifold, the 16-inch-long combustion chamber with manifolding ring for igniter oxygen. and the exhaust-nozzle assembly.

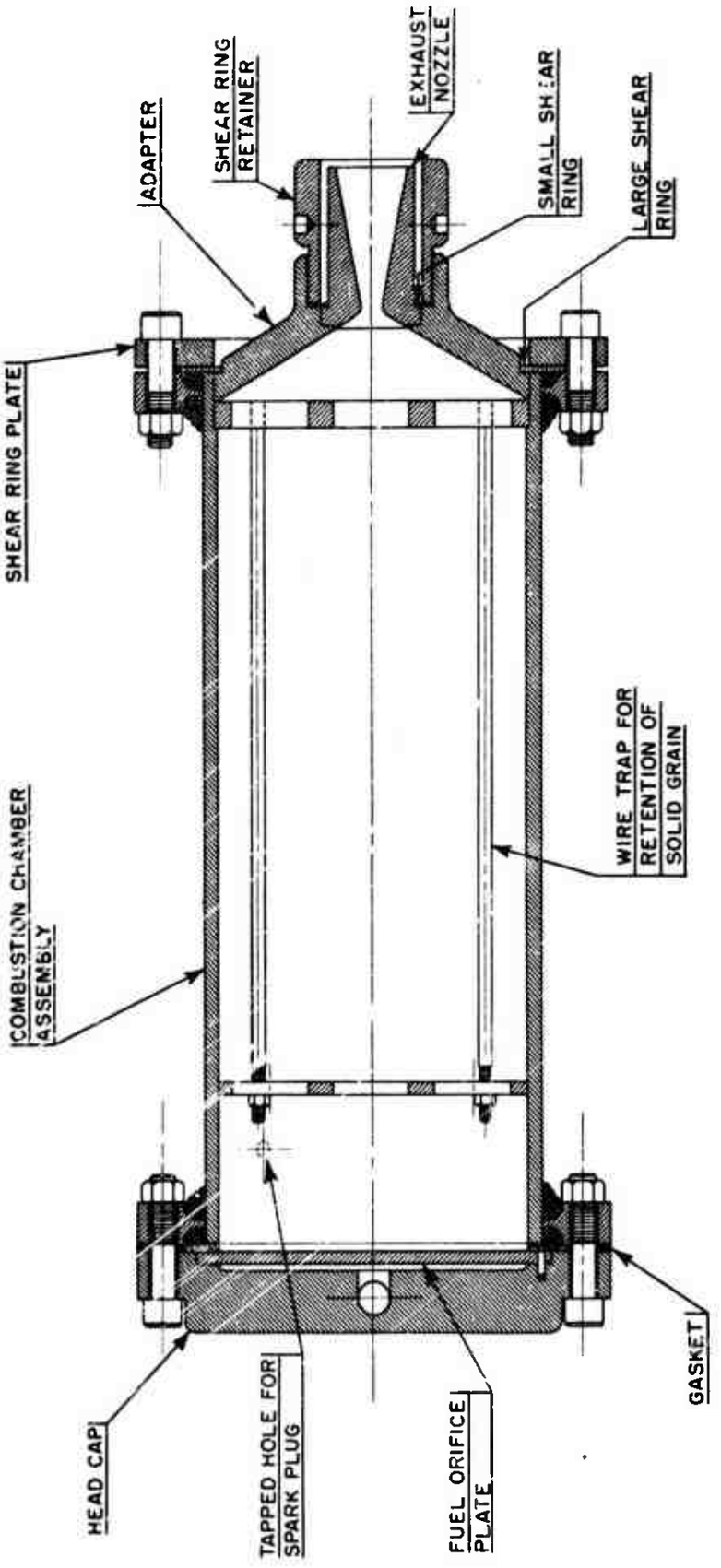


Fig. 6 LONGITUDINAL CROSS-SECTION DRAWING OF 6-INCH I.D. THRUST CHAMBER ASSEMBLY

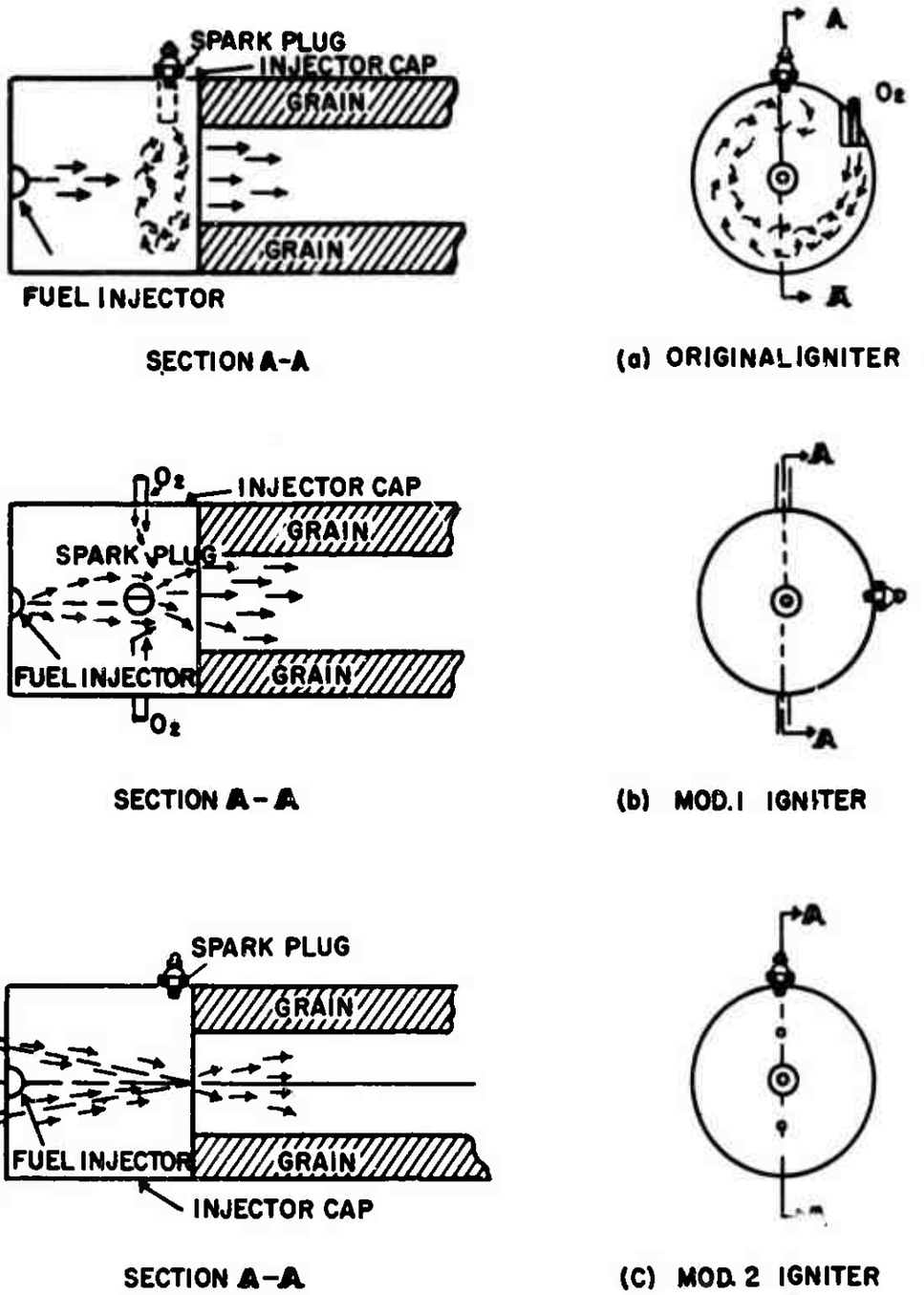
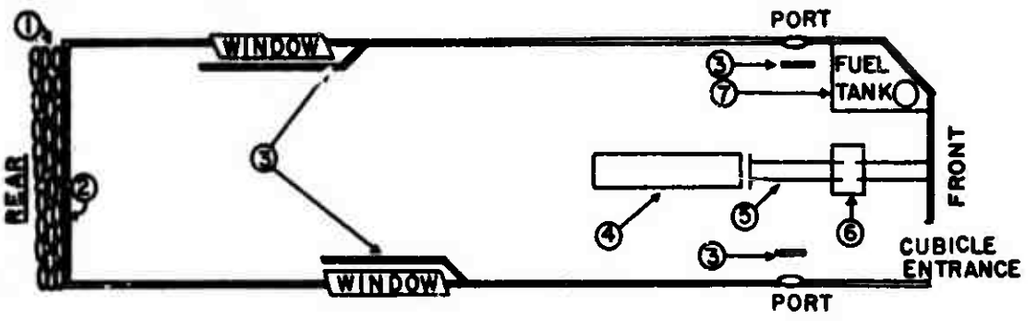


Fig. 7 THREE IGNITION-SYSTEM CONFIGURATIONS

A commercially available nozzle was used as the fuel injector, and served as the injector for both the igniter fuel and for the main operating fuel. A relatively low fuel rate was used for the operation of the ignition system. The fuel flow was subsequently increased to match the oxygen-evolution rate as the temperature of the oxidizer grain surface increased.

Tests on both the 2-inch and the 6-inch combustion chambers were conducted at the Forest Grove Burner Laboratory, Applied Physics Laboratory, The Johns Hopkins University. A fuel supply system (Ref. 21), a thrust stand, and data-recording apparatus were designed at APL for these tests. A converted nitrogen bottle served as a fuel tank and was used in conjunction with commercially available relief, check, and on-off solenoid valves. These components provided a safe and controllable high-pressure fuel system. The test installation is shown diagrammatically in Fig. 8. A schematic diagram of the fuel and ignition systems is given in Fig. 9. The ignition and operation controls were mounted on a single control panel which is shown in Fig. 10.

A mechanical-hydraulic system was employed to measure thrust. A cantilever provided the restraining force to the thrust of the operating rocket. The use of cantilevers of various sizes permitted a wide range of thrust measurement. In Fig. 11, a 2-inch thrust chamber is shown mounted in the thrust stand. Simultaneous recording of thrust, chamber pressure, and fuel flow rates was accomplished by the Bourdon type optical recording system (Ref. 24) shown in Fig. 12.



① 25 100-lb SAND BAGS; REINFORCEMENT OF 1/2-INCH DEFLECTOR PLATE
② 3/4-INCH PLATE; SANDBAG RETAINER
③ 1/2-INCH PLATE; WINDOW AND PORT PROTECTION
④ THRUST STAND
, 1 1/2-INCH ABUTTMENT PAD AND TWO PLATES; STABILIZE ROCKET AND TRANSFER THRUST TO CUBICLE WALL
⑥ 1/2-INCH PLATE; PROTECTION OF 12-INCH AIR LINE EXIT (NOT SHOWN)
⑦ 1/2-INCH PLATES; AROUND HIGH-PRESSURE FUEL TANK

Fig. 8 DIAGRAM OF LIQUID-SOLID ROCKET TEST FACILITY

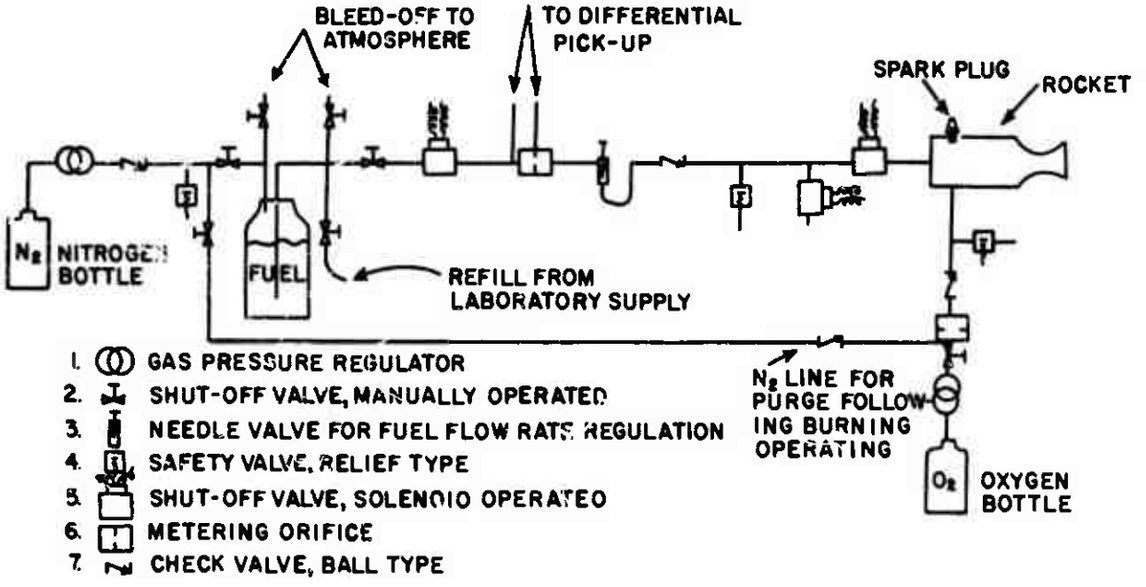


Fig. 9 FLOW DIAGRAM OF GAS-PRESSURIZED FUEL AND OXYGEN SYSTEMS FOR LIQUID-SOLID ROCKET

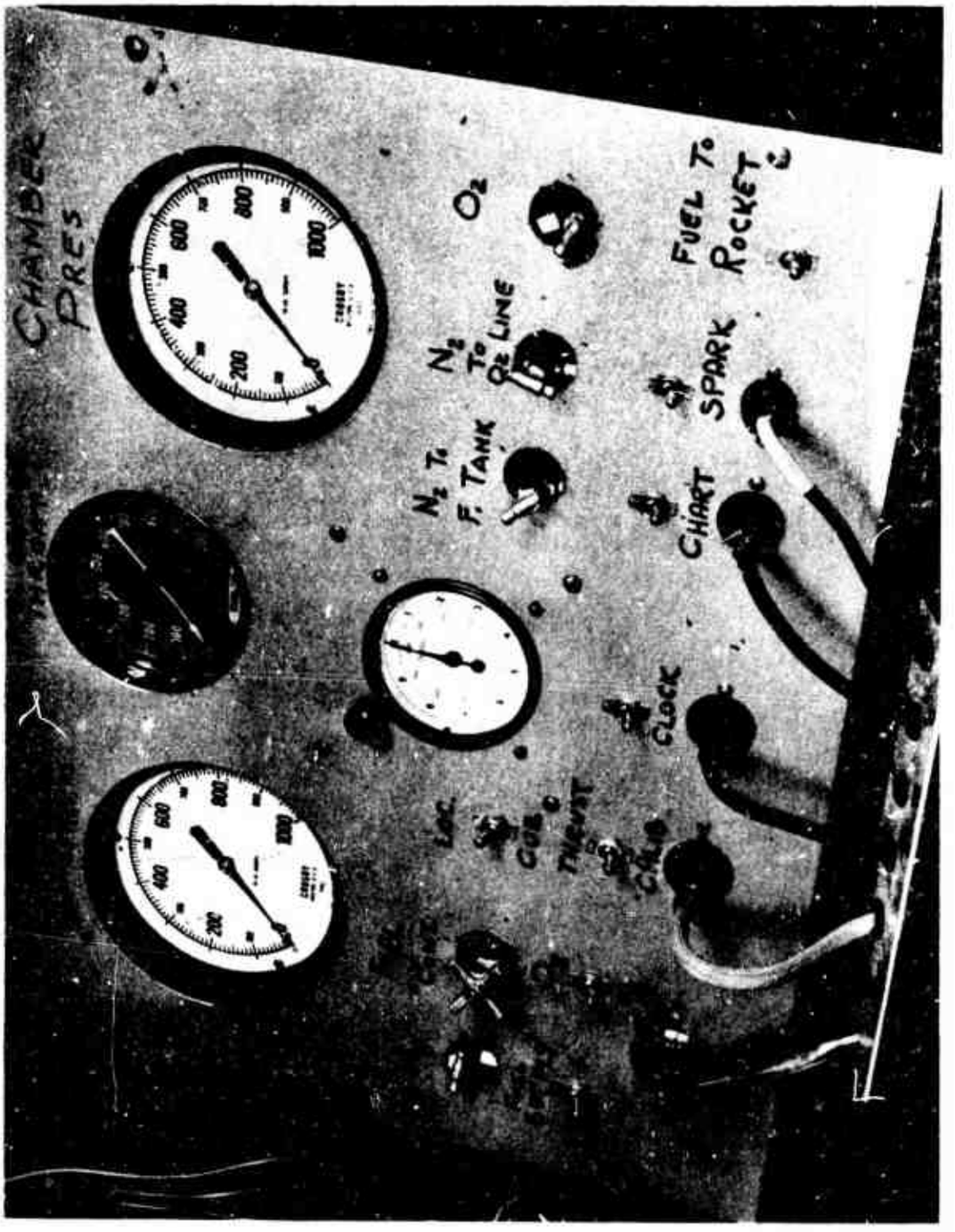


Fig. 10 CONTROL PANEL FOR TESTING OF LIQUID-SOLID ROCKET

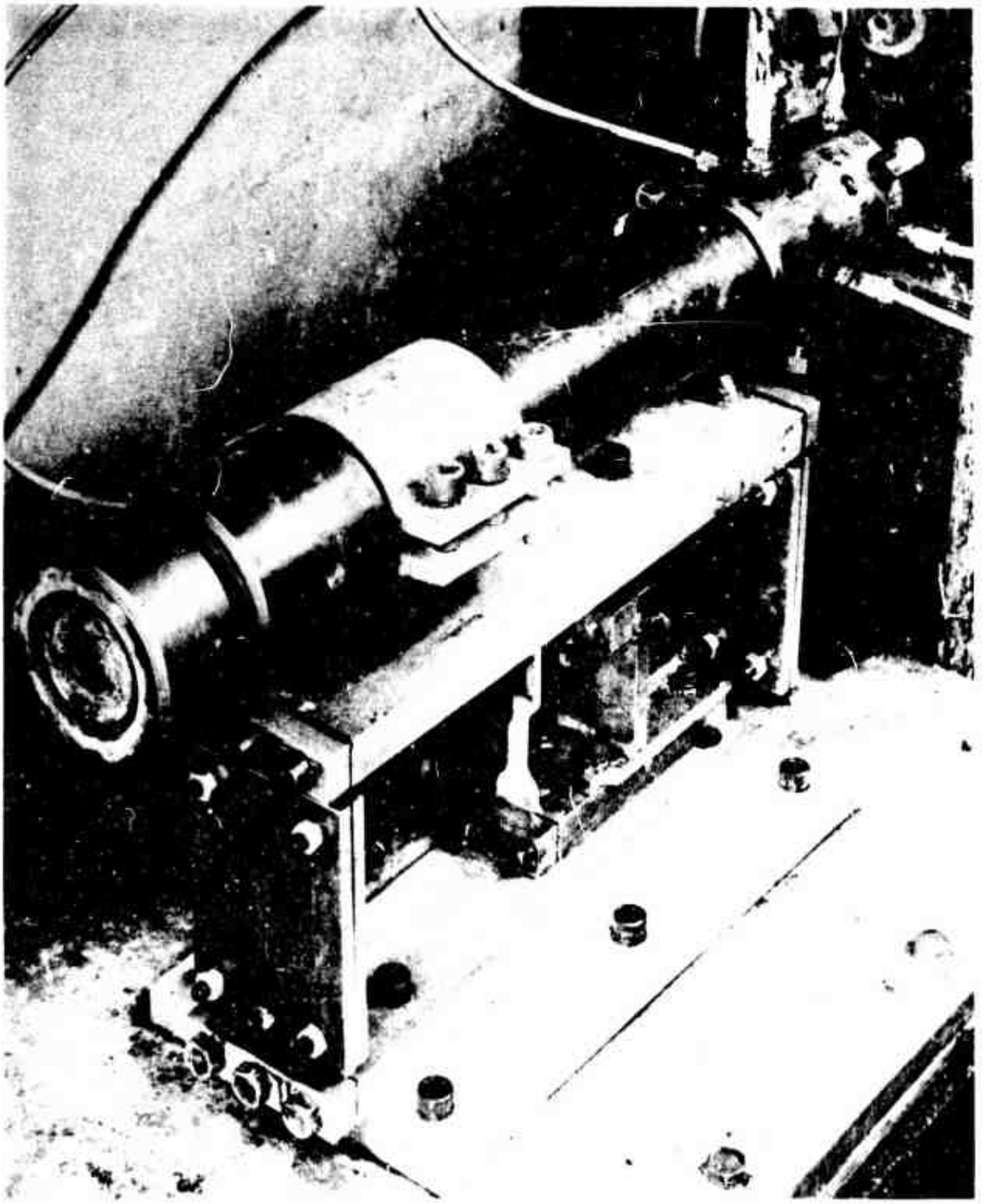


Fig. 11 TWO-INCH I.D. THRUST CHAMBER MOUNTED ON THRUST STAND

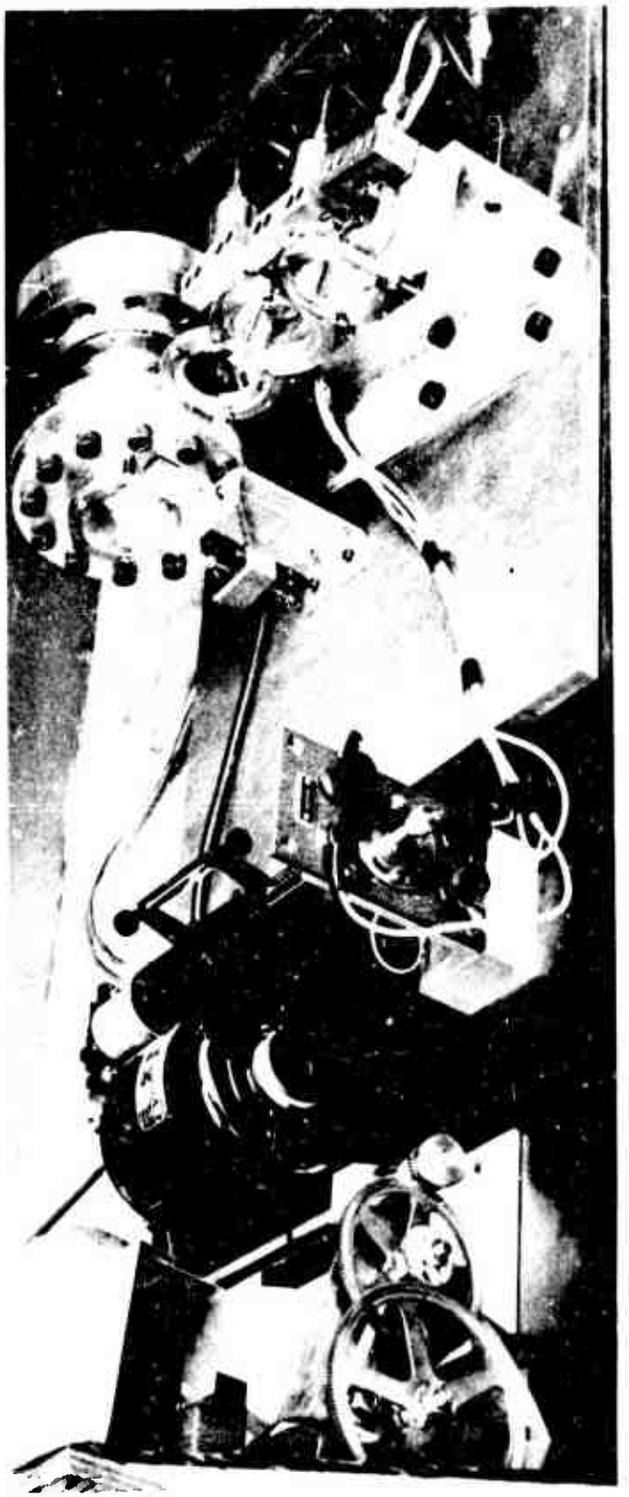


Fig. 12 OPTICAL-HYDRAULIC UNIT FOR RECORDING ROCKET FORCES AND FLOWS
The recording equipment is shown on the left. The Bourdon tubes are shown on the right.

IV. RESULTS AND DISCUSSION

Oxidizer Fabrication

The combustion properties of the three selected oxidizers, and of various combinations of the oxidizers, were determined by means of bench tests. The results of the determinations of combustion properties are tabulated in Table III. A list of the various compositions of the selected oxidizers which have so far been manufactured in the form of rocket grains is given in Table IV.

In an effort to discover the best method of manufacture for rocket grains, several different fabrication procedures were investigated. Table V contains a summary of both the methods and the results of the various fabrication procedures which were used for manufacturing Mod. 1 compositions.

Since the sensitivity of a material to detonation may serve as an index of the degree of safety with which it can be handled, standard impact tests (Refs. 25, 26) on the various oxidizers and oxidizer compositions proposed for use in the liquid-solid rocket were conducted at the Naval Ordnance Laboratory, White Oak, Maryland. The results of these sensitivity-to-detonation tests are summarized in Table VI. The values of TNT and tetryl are listed in order to provide a standard for comparison.

Rocket Testing

Much of the initial phase of the model testing program amounted to a search for operating procedures which would provide reproducible performance of at least fair quality. Systematic changes were made in the fuel injector and the fuel

TABLE III

Combustion Properties of Selected Oxidizers

Composition (Parts by weight)			Mod. Designation	Involves Oxygen	Yields Gaseous Products	Quenches without Fuel	Fuel Sustains Combustion	Catalyst	Binder	Remarks
$KClO_4$	NH_4NO_3	NH_4ClO_4								
Theoretical Ideal				Yes	Yes	Yes	Yes			
1	3	0	Mod.1, Mod.2	Yes	No	Yes	Yes	MnO_2	$NaNO_3$	Melts at 500° to 525°C and decomposes with binder and catalyst.
0	1	0	-	No	Yes (?)	Yes	No	NH_4^+ salts	$Mg(NO_3)_2 \cdot 6H_2O$	Melts at 170°C with binder.
0	0	1	-	No (?)	Yes	No	Yes	Fe^{++} salts	NH_4NO_3	Decomposes at <150°C; NO_2 fumes.
1	1	0	Mod.7	Yes (?)	No	Yes	Yes	MnO_2 and NH_4^+ salts		Melts at 100°C; decomposes at 110°C.
0	1	1	Mod.3, Mod.4	Yes	Yes	No	Yes Violet	Fe^{++} salts or NH_4^+ salts		Decomposes at <120°C.
1	0	1	-	Yes	No	Yes (?)	Yes Violet			Decomposes at ~150°C.
1	1	1	Mod.6	Yes (slow)	No	Yes Marginal	Yes	MnO_2 and NH_4^+ salts		Melts at 226°F.
1	2	4	Mod.5	Yes (slow)	No	No(?) Marginal	Yes	MnO_2 and Fe^{++} salts		Decomposes at ~150°C.

TABLE IV

Oxidizer Compositions

	Nitrate	Perchlorate	Catalyst	Other
Mod. 1	9% $NaNO_3$	89% $KClO_4$	2% MnO_2	Carbon
Mod. 2	8% KNO_3	83% $KClO_4$	2% MnO_2	7% $NaNO_2$
Mod. 3	47% NH_4NO_3	47% NH_4ClO_4	3% $(NH_4)_2Cr_2O_7$	3% $Mg(NO_3)_2 \cdot 6H_2O$
Mod. 4	48% NH_4NO_3	48% NH_4ClO_4	4% $Fe_2O_3 \cdot Fe_3O_4$	
Mod. 5	28% NH_4NO_3	56% NH_4ClO_4 14% $KClO_4$	2% Fe_2O_3, MnO_2	Carbon
Mod. 6	32.7% NH_4NO_3	32.7% NH_4ClO_4 32.7% $KClO_4$	1.1% MnO_2 0.4% $(NH_4)_2Cr_2O_7$	0.4% $Mg(NO_3)_2 \cdot 6H_2O$
Mod. 7	49.4% NH_4NO_3	49.4% $KClO_4$	0.6% MnO_2 0.6% $(NH_4)_2Cr_2O_7$	

TABLE V

Methods of Fabrication for $KClO_4$ Composition

Method	Temperature (°F)	Pressure (lb/in^2 gauge)	Time	Results	Remarks
Casting	590	1	3 hrs.	Excellent	For briquets
Compression:					
1) Motor mold	540	2000	4 hrs.	Good	Occasional grain fracture
2) Split mold	540	2000	4 hrs.	Poor	Grain cracks upon removal from mold
3) Strip mold	540	4000	4 hrs.	Fair	Occasional poor consolidation
4) Cold	Ambient	50,000	2-1/2 min.	Excellent	

TABLE VI

Sensitivity of Selected Oxidizers to Detonation

Chemical	50% Height (2.5 Kg)	With Catalyst	With Catalyst and Sugar
(TNT)	161 cm.		
Tetryl X64	42 cm.		
$KClO_4$	No explosions at 320 cm.	No explosion	320 cm. (1 explosion in 5 trials)
KNO_3	No	No	No
$NaClO_4$	No	No	No
$NaNO_3$	No	No	No
NH_4ClO_4	107 cm.	88 cm.	65 cm.
NH_4NC_3	No explosion	320 cm. (1 explosion in 6 trials)	320 cm. (1 explosion in 6 trials)
Mod. 1	No explosions at 320 cm.		
Mod. 6	No explosions at 320 cm.		
($KClO_4$ + NH_4NO_3)	No explosion	No explosion	
(NH_4ClO_4 + NH_4NO_3)	231 cm.	120 cm.	
(NH_4ClO_4 + $KClO_4$)	194 cm.		
(NH_4ClO_4, NH_4NO_3, $KClO_4$)		187 cm.	

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flow rate, and in the igniter and in the oxidizer composition so that the effect of such variations upon the chamber pressure, the linear burning rate, and the amount of grain burned might be observed.

When the minimal conditions of a standard operating procedure had been established, a series of more extensive tests was conducted. Representative data from the latter series of tests are presented in Table VII. It may be seen from Table VII that the test values of the specific impulse, I_{sp}, range from 64 to 110 lb-sec/lb for the Mod. 1 grain and from 112 to 130 lb-sec/lb for the Mod. 6 grain. Other workers (Refs. 27, 28) report values of I_{sp} of 180 and 187 lb-sec/lb for $KClO_4$ and NH_4NO_3 composite grains which may properly be compared with the above I_{sp} values for the Mod. 1 and the Mod. 6 grains, respectively. It should be recognized, however, that in composite grains the fuel and the oxidizer are intimately mixed at the time of fabrication (a condition necessary for optimum propellant performance), whereas in the liquid-solid rocket the mixing problem is still relatively unexplored.

Influence of Oxidizer Composition

The relative behavior of the different solid oxidizers investigated may be noted from the pressure-time curves shown in Fig. 13. The Mod. 1 grain is rather easily ignited, with an ignition delay of approximately 1 second, and yields the most stable combustion process. The Mod. 6 grain, on the other hand, is the most active. Mod. 7 appears to be quite stable also, although its chamber pressure is not so high as that of Mod. 1 and it is more difficult to ignite. Mod. 3 (not shown in Fig.13) was found to be highly unstable. The only test in which it was used resulted in a violent explosion. Mods. 2, 4, and 5 have not yet been tested operationally.

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TABLE VII

Summary of Representative Liqui

Test No.	Grain No.	Grain Length (inches)	Exit Nozzle Throat Diameter D_t (inches)	Area Ratio K	Total Oxidizer Burned W_o (lb)	Fuel Flow Rate $\dot{W}_f$ (lb/sec)	Residue % of Original Grain Remaining After Burning	Oxidizer to Fuel Rate Ratio W_o/W_f	Equivalence Ratio ø
Mod.1 Grain, 89% $KClO_4$, 9% $NaNO_3$, 2% MnO_2, Nominal K = 1000									
56	83	7	0.180	1050	0.344	0.0075	68	1.82	4.06
57	88	7	0.180	1050	0.5234	0.0080	30	2.0	3.7
58	90	7.25	0.180	1120	0.8125	0.008	25.5	3.4	2.2
63	95	6.50	0.180	890	0.578	0.125	41.0	1.55	4.8
73	106	5.125	0.150	1185	0.785		28.8		
Mod.1 Grain, Nominal K > 1000									
62	82, 86	10.50	0.190	1500	1.328	0.0145	25.5	3.85	1.92
71	73, 103	12	0.180	1850	1.583	0.0135	23.0	5.83	1.32
74	102, 105 109, 112	24	0.180	3350	1.203	0.0130	30.0	12.30	0.60
78	102, 105 109, 112	24	0.180	4880	1.375	0.0140	30.0	13.5	0.55
79	108, 110 114, 115	22	0.180	3100	3.785	0.0125	11.5	13.5	0.55
80	N_3, 111	8.75	0.180	1450	1.28	0.0103	21.8	4.5	1.85
81	N_2	11	0.180	3400	1.488	0.0135	26.0	8.8	0.833
Mod.6 Grain, 33% $KClO_4$, 33% NH_4NO_3, 33% NH_4ClO_4, Nominal K = 500									
76	50	5.125	0.250	425	0.50	0.0105	19.5	2.7	3.8
77	55	5	0.200	650	0.375	0.0112	19.2	2.8	3.6
Mod.7 Grain, 50% $KClO_4$, 50% NH_4NO_3, Nominal K = 1000									
58	81	7.25	0.180	1120	0.53	0.0080	3.5	2.14	5.8

VII

Liquid-Solid Rocket Tests

Equivalence Ratio ø	Burning Time t_b (sec)	Grain Linear Burning Rate r (in/sec)	Chamber Pressure $\bar{P}_c$ (lb/in^2 gauge)	Thrust $\bar{F}$ (lb)	Specific Impulse $\bar{I}_{sp}$ (lb-sec/lb)	Remarks
4.06	20	0.025	136			F not measured. Inferior fuel injection caueed quenching.
3.7	25.5	0.0195	155			F not measured.
2.2	30.5	0.0164	130			F not measured.
4.8	25	0.0200	140			F not measured.
	23	0.0219	310			F not measured. Fuel differential pick-up not functioning.
1.92	24	0.021	240			F not measured. Fuel flow measured visually.
1.32	19	0.031	364			F not measured.
0.60	4.5	0.055	539	19	64	F estimated. Value of F beyond limit of sensing element.
0.55	4.5	0.055	974	19	71	Reignition of grain remaining from Test No. 74.
0.55	12.7	0.039	585	21	96	
1.95	27.0	0.0195	240	9	99	2% carbon added to Mod.1 grain.
0.933	11.5	0.0430	445	14.3	110	Grain consists of 11 pellets each 1" thick.
3.9	19.0	0.029	195	13.0	112	
3.9	13.5	0.037	322	11.5	130	
5.8	32.0	0.0155	125			F not measured.

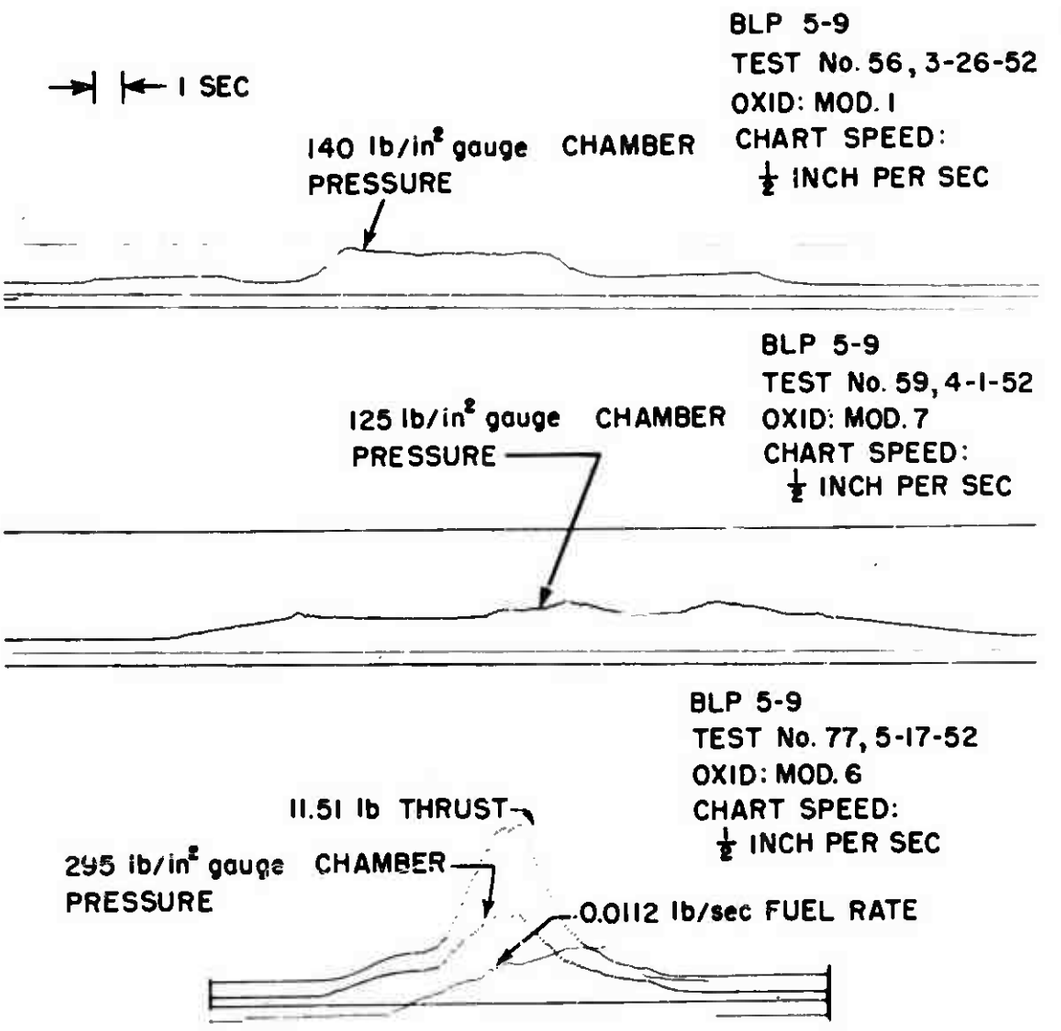

Fig. 13 CHAMBER PRESSURES WITH VARIOUS OXIDIZER COMPOSITIONS

These three tests were conducted in 2-inch I.D. combustion chambers which contained grains 6 inches long and which utilized exhaust nozzles of 0.180-inch throat diameter. In Test No. 56 and in Test No. 59 the fuel rate value exceeded the limit of the recording equipment, and is not shown.

Table VII shows also that, for approximately equal amounts of grain, the Mod. 6 composition, with a nominal ratio of grain burning surface area to nozzle throat area, K, of 500, yields a chamber pressure, specific impulse and burning rate higher than the Mod. 1 composition with a nominal K of 1000. Moreover, one test using Mod. 6 grain (Test No. 77) shows that the chamber pressure was still increasing at the time of grain depletion and that it had already attained a value of 295 lb/in.^{2} gauge. This failure to attain steady-state operation had previously been noticed in the testing of thin webs and short grain lengths. Increased performance is expected with longer grain lengths, thicker webs, and higher K's.

A point of caution which should be noted regarding the use of Mod. 6 grain concerns its potential destructiveness if not properly ignited. Drop test data do not confirm this point (see Table V), but it was found that if unburned igniter fuel became impregnated in a poorly consolidated grain, an explosion could result upon eventual ignition.

A further evaluation of the various grain compositions was based on the per cent of oxidizer which was consumed during the burning operation. Although the Mod. 1 composition consists of only 50 per cent oxygen, it was usually found that only 25 per cent of the grain remained as residue. This implies that combustion-chamber gas temperatures are sufficiently high to vaporize a portion of the diluent grain solids. Theoretically, the Mod. 6 composition should result in less residue than the Mod. 1 composition, since the former contains a larger percentage of oxygen than the latter. This is tentatively confirmed by the 19 per cent residue for the Mod. 6 composition reported in Table VII.

Influence of Area Ratio, K

It is apparent from Table VII that increasing the length of the Mod. 1 oxidizer grain from 7 to 22 inches results in an increase of the chamber pressure from 138 to 585 lb/in.^{2} gauge. The pressure-time curves of Tests No. 56, 62, and 79 presented in Fig. 14 further demonstrate this point.

When the results of the tests mentioned above are considered with the results of Test No. 81, it becomes evident that the area ratio, K, was the significant factor which contributed toward increasing the performance. This is shown clearly in the data tabulated below, which have been extracted from Table VII.

Performance at Constant Grain L, and Variable K

Test No.	Grain L (inches)	K	P_c (lb/in.^{2} gauge)	I_{sp} lb-sec/lb
62	10.5	1500	240	
81	11	3400	445	110

Performance at Variable Grain L, and Constant K

Test No.	Grain L (inches)	K	P_c (lb/in.^{2} gauge)	I_{sp} lb-sec/lb
79	22	3100	585	96
81	11	3400	445	110

Consequently, variations in grain length should be regarded only as one of the mechanisms by which K may be varied.

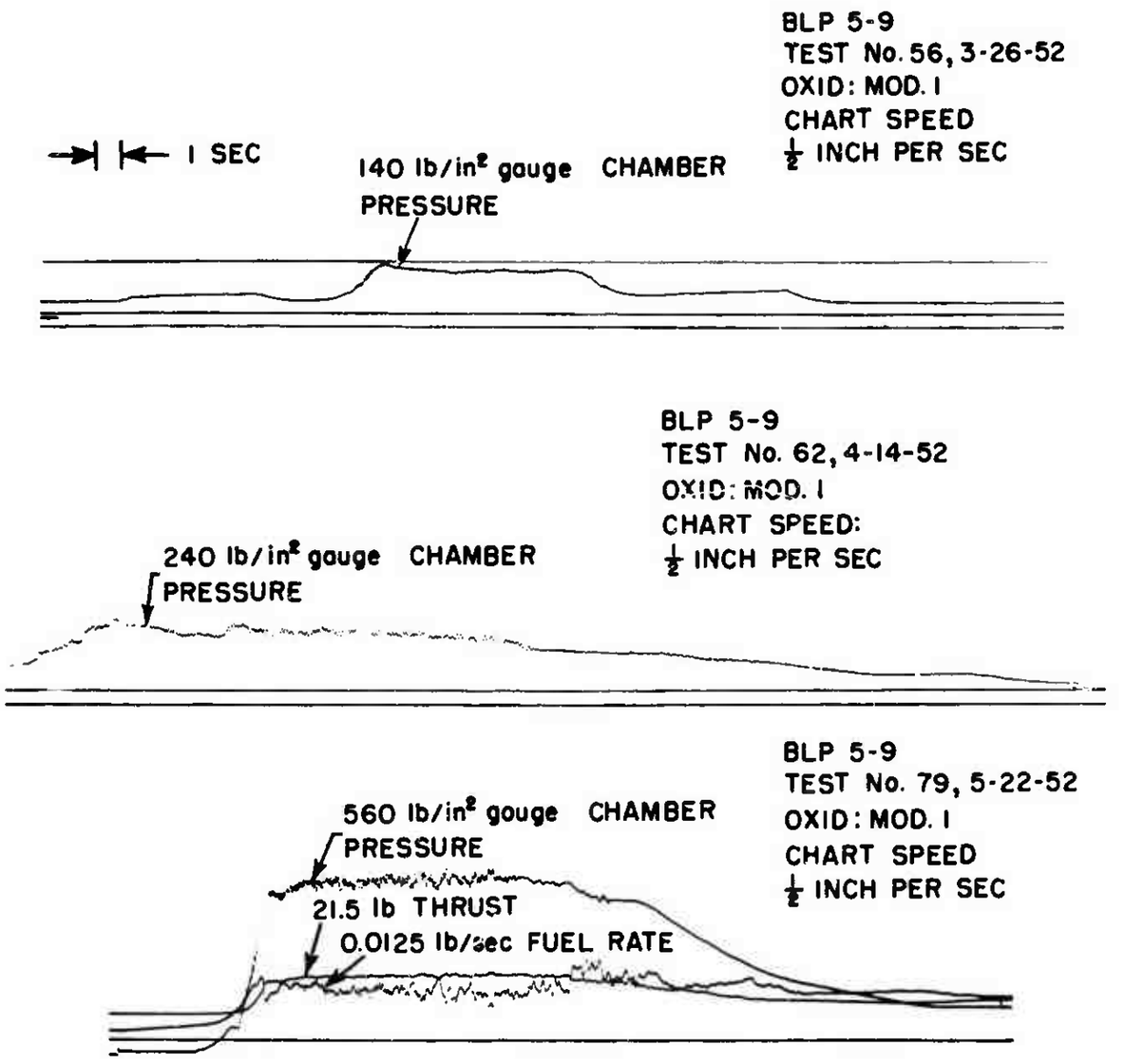

Fig. 14 EFFECT OF GRAIN LENGTH ON CHAMBER PRESSURE

The grain cross section and the exhaust-nozzle throat diameter were the same in these three tests. The grain length was 7 inches for Test No. 56, 12 inches for Test No. 62 and 22 inches for Test No. 79. In Test No. 56 the fuel rate value exceeded the limit of the recording equipment and is not shown. In Test No. 62 the fuel rate was not recorded.

Tests of various grain lengths also permit a comparison of the change in the linear burning rate, r, of the grain with the combustion chamber pressure, P_c. If the solid-propellant burning-rate theory, in which $r = BP_c^n$, is assumed, then experimental results indicate that $n = 0.622$ (Fig. 15) for the Mod. 1 oxidizer composition as compared with $n = 0.75$ (Fig. 16) for a $KClO_4$ composite grain (Ref. 27). Table VII shows, however, that usually an increased fuel flow rate accompanies the increase of r with P_c. Consequently, the r versus P_c function at a constant fuel flow rate still remains to be established.

Influence of Exhaust Nozzle Throat Area

It may be seen in Fig. 17 that the chamber pressure increases from 138 to 310 lb/in.^{2} gauge when the nozzle throat diameter is decreased from 0.180 to 0.150 inch at constant K for the Mod. 1 grain. This 30 per cent decrease in nozzle throat area in Mod. 1 is accompanied by a 9 per cent increase in r; whereas for the Mod. 6 grain, a comparable area ratio change results in a 55 per cent increase in r.

Chamber Pressure Fluctuations

The addition of 2 per cent carbon to the Mod. 1 grain used in Test No. 80 produced an exceptionally smooth chamber pressure, as shown in Fig. 18. As an example of the Mod. 1 grain performance without the carbon, the pressure-time curve for Test No. 81 is included for comparison in the same figure.

Controllability

The controllability of the liquid-solid rocket and its response to a stoppage of fuel flow was demonstrated in Tests No. 78

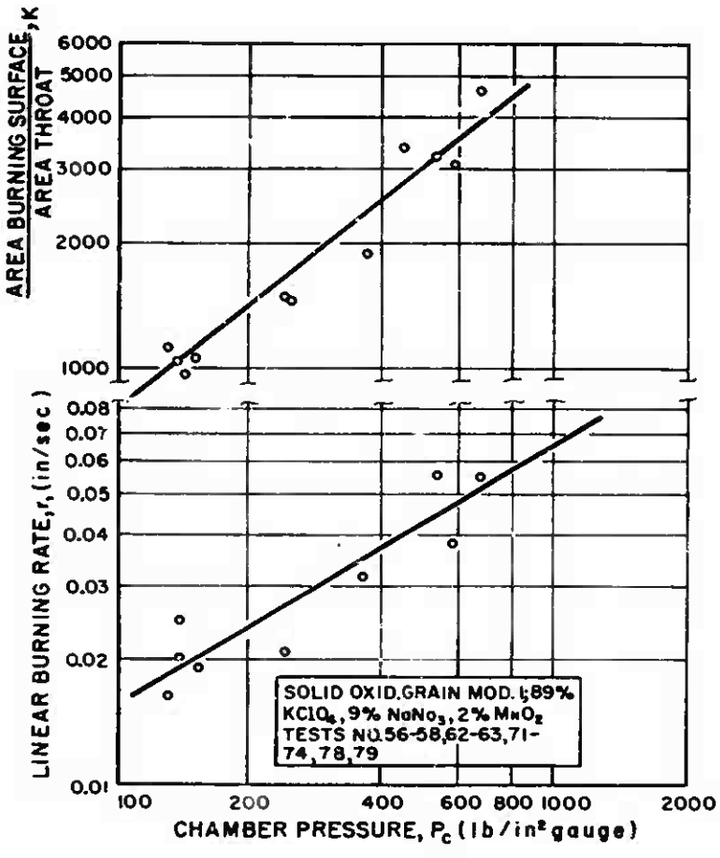

Fig. 15 GRAIN BURNING PROPERTIES

The straight lines on this graph are accurate under steady-state conditions only.

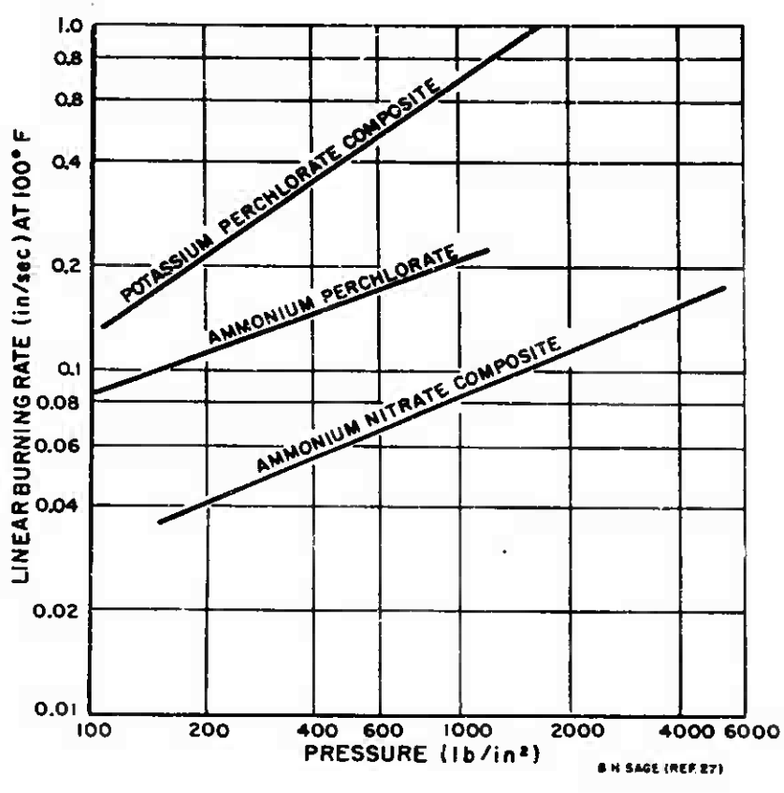

Fig. 16 LINEAR BURNING RATES FOR REPRESENTATIVE SOLID COMPOSITE PROPELLANTS

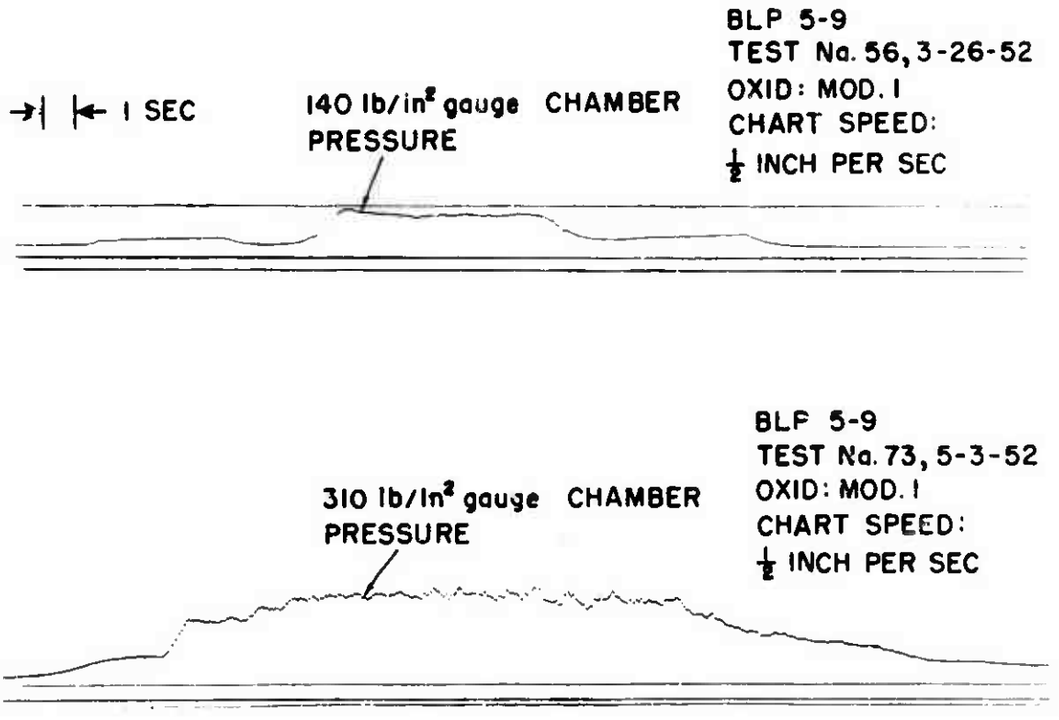

Fig. 17 EFFECT OF EXHAUST-NOZZLE THROAT AREA ON CHAMBER PRESSURE

The exhaust-nozzle throat diameter was 0.180 inch for Test No. 56 and was 0.150 inch for Test No. 73.

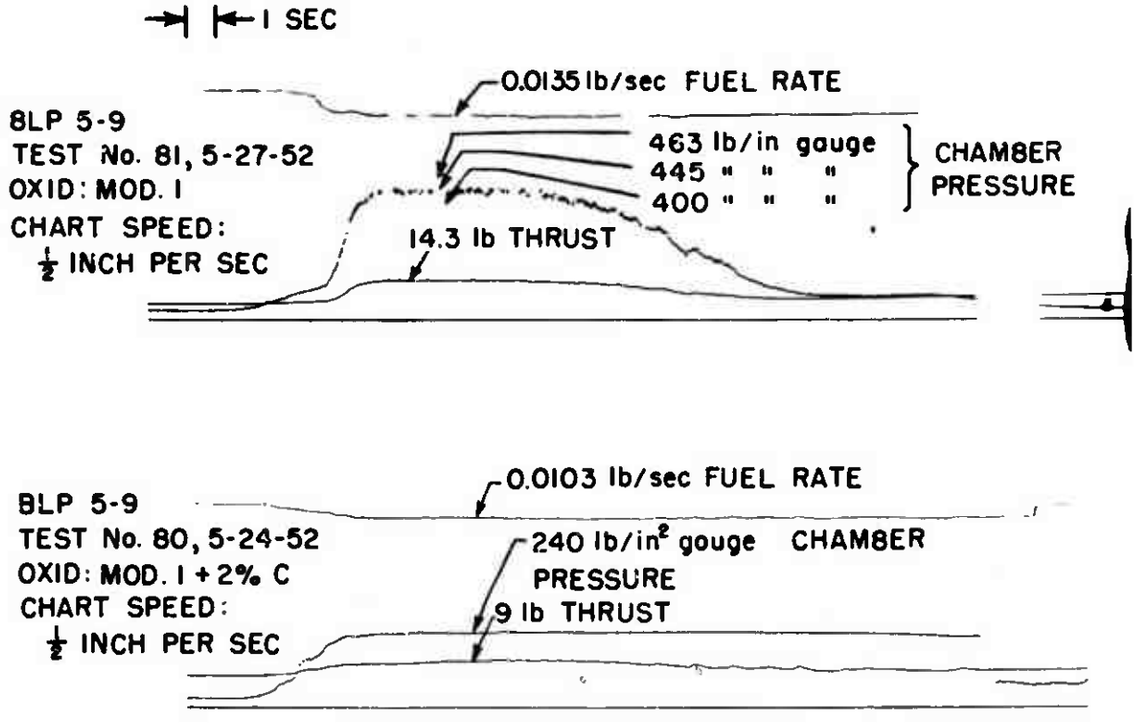

Fig. 18 STABILIZING EFFECT OF CATALYST ON CHAMBER PRESSURE

Two per cent carbon was added to Mod. 1 oxidizer composition for Test No. 80.

and 79 in which the identical grain was used. The chamber pressure fell to zero in 3 seconds following the shut-off of fuel at the end of Test No. 78. Reignition was obtained 1/2 second after the fuel flow was turned on again in Test No. 79.

V. CONCLUSIONS

This program has served to demonstrate the feasibility of a liquid-solid rocket which uses a propellant combination of liquid hydrocarbon fuel and solid inorganic oxidizers.

From the results stated in Chapter IV, the following specific conclusions can be drawn:

Oxidizer Fabrication

Compositions

Several oxidizer compositions using potassium perchlorate, ammonium nitrate, and ammonium perchlorate have been successfully fabricated and fired in a liquid-solid rocket. Each composition presents its own problem of ignition, fuel injection, and testing.

A small addition of the proper catalyst can be used to increase the rate of oxidizer decomposition and to reduce the amplitude of chamber-pressure fluctuations.

Fabrication Methods

Excellent fabrication of oxidizer grains can be accomplished safely by high-pressure, cold-compression molding without the use of binding agents. This method could provide a grain of uniformly high quality at a high rate of production with the use of a minimum number of personnel.

Sensitivity

All compositions with the exception of those containing a large proportion of ammonium perchlorate have been found to be stable and insensitive to detonation tests. Thus, the oxidizers for liquid-solid rocket grains can be made and stored under standard commercial manufacturing conditions. The non-explosive ingredients provide a safe, easily handled, rocket grain.

Rocket Testing

Elementary solid and liquid rocket theory has so far been adequate for the design and testing of the liquid-solid rocket.

Compositions

Mod. 1 grain is safe and easily ignitible. Ignition procedures for other compositions require further development.

Rocket Variables

Limited developmental testing in test chambers where heat losses are high has yielded specific impulse values for the Mod. 1 grain of 50 per cent, and for the Mod. 6 grain of 70 per cent of specific impulse values obtained in experiments with composite solid rocket grains in which the fuel-oxidizer ratio was nearly stoichiometric.

Controllability

Operational control and flexibility has been demonstrated by causing variable discharge rates of the combustion gases by means of variations both of the amount of grain present and of the fuel injection rate. Grain burning may be quenched by fuel shut-off and then successfully reignited.

VI. RECOMMENDATIONS

On the basis of the findings herein reported it has been recommended that a broader program for the complete and successful development of the liquid-solid rocket be activated.

The aim of the program will be to develop a full-scale JATO unit using a liquid hydrocarbon fuel and a solid oxidizer, with emphasis placed on safety of manufacture and handling; on the use of readily available, inexpensive materials; and on the use of commercial scurces of supply.

The specific aspects of the program which remain unexplored consist primarily of problems connected with the optimization of the propellant system. The solution of these problems will involve oxidizer studies, more extensive testing of rocket performance, and both theoretical and experimental studies of scaled-up rocket design. The oxidizer studies will include investigation of grain compositions, grain combustion properties, and methods of grain fabrication. Small-scale and large-scale experimental tests will provide data which will assist in the optimization of the ignition and fuel injection systems and which will be used in a further evaluation of over-all rocket performance. The aim of the studies in rocket design will be to determine scaling parameters for the production of a full-scale JATO unit. In addition, studies will be made of the factors which affect the safe manufacture and handling of rocket components, and of means for producing components of consistently high quality.

The facilities required for the completion of this program include such items as a high-pressure compression molding press; weighing, grinding, and mixing equipment for chemicals; quality control equipment; safety testing facilities; and installations for scaled-up performance testing.

As a result of the above recommendations, a development program leading toward the production of a full-scale liquid-solid JATO (1000-pound thrust) unit is being continued at the Applied Physics Laboratory.

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